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ENGINEERING REPORT NO. 474.5

FC

DERIVATION OF WEIGHT R_F TERMS OF
PARAMETRIC DESIGN ANALYSIS FOR
PROPELLOPLANE TRANSPORT STUDY
Contract Nonr 1657(00)

[redacted]

474.5

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HILLER HELICOPTERS

PALO ALTO, CALIFORNIA

ENGINEERING REPORT

REPORT NO. 1048

MODEL NO. 474.5

TITLE Derivation of Weight RF Terms of Parametric Design Analysis for Propelloplane Transport Study Contract Nonr 1657 (00)

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This document has been reviewed in accordance with OPNAVINST 5510.17, paragraph 5. The security classification assigned hereto is correct.

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LIST OF SYMBOLS

A_p - Propeller Disc Area, ft^2

AF - Propeller Blade Activity Factor

A_q - IxP Propeller Excitation Factor

AR - Aspect Ratio

B - Number of Propeller Blades

b - Wing Span, Feet

c - Wing Chord, Feet

D_p - Propeller Diameter, Feet

HP - Installed Normal Rated Horsepower, Standard Day, Sea Level

K_p - Ratio of HP/Thrust

M_O - IxP Propeller Blade Root Bending Moment, Ft.-Lb.

R - Ratio of Weight to Aircraft Design Gross Weight

Q - Torque, Ft-Lb.

S - Wing or Tail Surface Area, ft^2

V_T - Propeller Tip Speed Ft./Sec.

W - Weight

w_H - Hover Disc Loading, Defined as Hover Thrust/A_p, Lb./ft^2

W/S - Wing Loading, Design Weight/Wing Area, Lb./ft^2

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SUMMARY

This report summarizes the structural design criteria and presents the derivation of the weight R_F equation for parametric determination of the design parameters of the minimum gross weight aircraft capable of fulfilling the performance specifications of Contract Nonr 1657 (00).

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INTRODUCTION

The Weight R_F Equation

Use of the Hiller R_F method of parametric optimization for the specified transport propelloplane mission requires the development of an analytical expression for the variation of the ratio of fuel weight to gross weight which is permissable at any gross weight in terms of the var_ .bles of the investigation.

In general

$$W_g = W_P + W_C + W_E + W_F + W_{FT}$$

where

W_g = Gross Weight $\quad$ W_P = Design Payload

W_C = Crew Weight $\quad$ W_E = Empty Weight Less Fuel Tanks

W_F = Allowable Fuel Weight $\quad$ W_{FT} = Fuel System Weight

re-writing

$$W_F + W_{FT} = W_g - W_P - W_C - W_E$$

and placing into ratio form by dividing by gross weight

$$R_F + R_{FT} = 1 - R_P - R_C - \Phi$$

where Φ is designated as the ratio of empty weight, less fuel system weight, to gross weight. To ensure compatibility between fuel and fuel tank weight, tank weight is assumed to be proportional to the amount of fuel stored, hence to fuel weight.

$$R_F (1 + k_F) = 1 - R_P - R_C - \Phi$$

Therefore, the R_F can be expressed

$$R_F = \frac{1}{(1 + k_F)} (1 - R_P - R_C - \Phi)$$

This equation is the generalized weight equation of the aircraft and is referred to as the <u>Weight R_F Equation</u>. The R_F parameter provides the common link between weight and aerodynamic characteristics.

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For the mission of this contract, the design payload is specified as 8000 lbs., a crew of three weighs 600 lbs., and the weight of self sealing tanks is assumed to be .9 lbs. per gallon of fuel.

Thus

$$R_P = 8000/W_g$$

$$R_C = 600/W_g$$

$$1 + k_F = (1 + \frac{.9}{6}) = 1.15$$

hence

$$R_F = \frac{1}{1.15}\left(1 - \frac{8600}{W_g} - \Phi\right)$$

The remaining unknown, Φ, is the sum of the weight ratio expressions of all individual components which comprise the empty weight of the aircraft and is, therefore, a function of the design parameters affecting each.

Five parameters are chosen as the fundamental variables of the study. These are:

1. Disc Loading
2. Tip Speed
3. Aspect Ratio
4. Wing Loading
5. Gross Weight

The major effort of the weight analysis is, therefore, the derivation of the weight expressions for the component items of Φ in terms of these five variables.

Structural Criteria and Weight Prediction Approach

By nature of the broad scope of the parametric analysis utilized in this study, establishment of structural design criteria is limited to generalizations sufficient to insure realistic weight estimations of the aircraft components whose weights are a function, in some manner, of the aircraft loads.

Design loading for propeller blades is established to be the more critical of the IxP vibratory moments occurring during transition or normal fixed wing flight regimes.

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Airframe and wings are designed to +5.0, -3.5g ultimate load factors in order to provide a general strength level adequately representative of symmetrical maneuver and landing conditions for aircraft of this size and function.

Design loading for the wing tilting mechanism, which is most critical during an asymmetrically braked forward landing roll with wing tilted to vertical position, is approximated by an equivalent 2.63g load factor applied forward through c.g. of hinged mass when the wing is vertically positioned.

The approach to the problem of practical weight prediction considers the fact that, in general, design requirements for most of the aircraft components are similar to those of current conventional aircraft, and the weights of these components can be most practically expressed by empirical equations derived from data on operational aircraft with similar design parameters. These weights are quite representative of current design practice. Wherever required for components which are peculiar to this type of aircraft, or represent unique applications of conventional components, detailed treatment on a more analytical basis is accorded.

DERIVATION OF THE Φ TERM

The components comprising the empty weight items of this aircraft are divided into the following nine groups.

A. Power Plant (Including engines, transmissions, engine controls, accessories, engine mounts, vibration dampers, nacelle.)

B. Propellers (Excluding propeller controls, anti-icing and spinners.)

C. Wings

D. Fuselage

E. Landing Gear

F. Empennage

G. Supplemental control system (includes auxiliary engines, ducts, jet deflecters, fuel, etc.)

H. Wing tilting mechanism

I. Fixed and Operational Equipment (Includes surface control systems, hydraulic, electrical, pneumatic systems, furnishings, navigation equipment, anti-icing, and air conditioning provisions, electronics, etc.)

Expressions for each group are derived individually below, and the sum of the expressions defines Φ.

A. Power Plant Weight Ratio

Power Plant Weight includes the weight of the engines, transmissions, engine controls and accessories, engine and transmission oil, and oil systems, engine mounts, vibration dampers, firewalls, and nacelle cowling.

The gas turbine engines for this aircraft are similar in construction to the current Allison T-40 engine, in that there are two power sections geared to a common transmission in each nacelle. From the generalized specific weight curves for engines forecast for 1965 (Reference 1), the following relationship between engine weight and normal rated power at sea level is derived for the range of horsepowers indicated.

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$$\left(\frac{W}{HP}\right)_{nacelle} = 2.02\ HP_{nacelle}^{-.26} \quad (3000 \leq HP_{nacelle} \leq 11,000)$$

hence, the weight ratio of the engine in one nacelle is

$$\frac{W_e}{W_g} = \frac{2.02\ (HP^{-.26})HP}{W_g} = \frac{2.02\ HP_{nacelle}^{.74}}{W_g}$$

Total normal rated design thrust at sea level, as required for hover ceiling requirements, = T = 1.3 W_g, so the HP of one nacelle of a four nacelle aircraft may be expressed

$$HP_{nacelle} = \frac{1}{4}\ HP_{tot} = \frac{1}{4}\left(\frac{HP}{T}\right)T = \frac{1.3}{4}\left(\frac{HP}{T}\right)W_g = \frac{1.3}{4}\ K_p\ W_g$$

where K_p = HP/T and HP = total normal rated horespower at sea level installed in the aircraft.

Hence, the engine weight ratio may finally be written

$$\frac{W_e}{W_g} = \frac{2.02}{W_g}\left[\frac{1.3}{4}\ K_p\ W_g\right]^{.74} = .88\ K_p^{.74}\ W_g^{-.26}$$

This expression includes weight of accessories and engine controls, but not reduction gearing.

Transmissions for this aircraft couple the two power sections of each nacelle to the propeller shaft and are of the planetary train type with two inputs and a coaxial output. Statistical data for transmissions of this type, which includes the weights of gearboxes and centrifugal and overrunning clutches, indicate the following relationship between transmission weight and maximum torque on the low speed output (References 2 and 3).

$$W_T = .081\ k_n\ Q^{.88}$$

k_n, a factor to account for the number of inputs and outputs is evaluated to be 1.40 from studies of current transmissions of this type (T-54, T-56, T-40).

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For one nacelle, total design torque is assumed to be 75 percent maximum torque available at sea level military power. This derating of the transmission effects a significant weight saving in view of the large excess of power necessarily available at sea level in order to meet the hover requirements of 6000 feet on a 95° F day.

thus

$$Q = .75 \frac{550 \text{ HP DP}}{2V_T}$$

Using the previous notation for horsepower/nacelle, military horsepower may be expressed

$$HP_{MIL} = 1.12\ HP_{NA} = 1.12\left(\frac{1.3}{4} K_P W_g\right)$$

and defining the hover disc loading as T/total disc area

$$D_p = 2\left(\frac{1.3\ W_g}{4\ \ w_H}\right)^{\frac{1}{2}}$$

Hence, the weight ratio of one transmission is

$$\frac{W_T}{W_g} = 3.45\left(\frac{K_P}{V_T}\right)^{.88} \frac{W_g^{.32}}{w_H^{.44}}$$

Oil consumption for engines and transmissions is conservatively assumed to be 1.5 gallons per hour per nacelle, based on average requirements of present day engines. Oil tanks weigh approximately two pounds per gallon of oil, and turbine grade oil weight is assumed to be 7.9 lb/gallon.

Hence, the weight ratio of oil and oil system per nacelle may be expressed, for an assumed three hour mission with 100 percent reserve

$$\frac{W_{oil\ +\ oil\ sys.}}{W_g} = \frac{(1.5)6(7.9 + 2)}{W_g} \quad \frac{70 + 20}{W_g} = \frac{90}{W_g}$$

Starter weight is neglected since the engines are started by the auxiliary power unit bleed air.

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The nacelle weight includes engine mounts, oil cooling systems, firewalls, vibration isolation systems, cowlings, etc. Present day installation weight averages about 68 percent of engine weight. Assuming that the twinned engine system increases installation weight required for a single engine by 50 percent, then

$$\frac{W_N}{W_g} = \frac{3}{2}\left(\frac{.68}{2}\right)\frac{W_e}{W_g} = .51\frac{W_e}{W_g}$$

or

$$\frac{W_N}{W_g} = .45\ K_P^{.74}\ W_g^{-.26}$$

Total power plant weight ratio is the sum of the above expressions (one nacelle only).

$$\frac{W_{PP}}{W_g} = 1.33\ K_P^{.74}\ W_g^{-.26} + \frac{90}{W_g} + 3.45\left(\frac{K_P}{V_T}\right)^{.88}\left(\frac{W_g}{V_T^{.44}}\right)^{.32}$$

K_P, the ratio of total installed normal rated horsepower at sea level to design thrust is a characteristic of the propellers chosen and will be related to tip speed and disc loading in section B.

The above expression, of course, represents "rubber engines". For those portions of the study where hardware engines are required, weights are taken from the appropriate engine specifications.

B. Propeller Weight Ratio

Due to the severe vibratory loadings to which propellers for VTO aircraft are subject, propeller blade weight is best described in terms of the IxP loadings as follows: (Reference 4)

$$W_B = .0536\ \frac{M_O}{D_P^{.70}} + .000844\ AF\ D_P^{2.3}$$

M_O is the critical IxP moment at the blade zero station. Thickness to width ratio at the .4 radius is approximately .148.

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Weight of the propeller hub for both single and dual rotation propellers is approximately 50 percent of total propeller weight for the rapid pitch change rates required with turbine engines. Hence, the weight ratio of the entire propeller may be written

$$\frac{W_P}{W_g} = \frac{1}{W_g}\left(BW_B + W_{HUB}\right) = \frac{2B}{W_g} W_B$$

$$= \frac{2B}{W_g}\left[.0536 \frac{M_O}{D_P^{.70}} + .000844 \, AF D_P^{2.3}\right]$$

Defining hover disc loading as in the previous section, propeller diameter may be expressed

then

$$D_P = 2\left(\frac{1.3\, W_g}{4\pi\, w_H}\right)^{\frac{1}{2}}$$

and total weight ratio of one propeller is

$$\frac{W_P}{W_g} = .1455 \left(\frac{B(AF)\, W_g^{.50}}{w_H^{1.5}}\right)\left(\frac{M_O\, w_H^{1.5}}{AF\, W_g^{1.5}} + .00419\right)$$

This expression is equally valid for single and dual rotation propellers, since, in practice, D and AF are the same for forward and aft blades, the relationship between hub and blade weight is similar for both types, the B term accounts for the actual number of blades, and M_O may be assumed equal for forward and aft blades. Although calculations would show M_O smaller for aft blades due to straightening of the inflow, interference buffeting removes much of the conservatism of this assumption.

For simplification of calculations, the parameter M_O^* is used in place of M_O

$$M_O^* = \frac{M_O}{\frac{2}{B}\left(\frac{W_g^3}{w_H}\right)^{\frac{1}{2}}}$$

Hence

$$\frac{W_P}{W_g} = .291\left(\frac{W_g}{w_H}\right)^{.15}\left[M_O^* + .0021 \frac{B(AF)}{w_H}\right]$$

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In this expression M_O^* and B(AF) are not fundamental parameters of the R_F equation, but M_O^* in the critical conditions can be related to disc loading and tip speed, and B(AF) will be optimized by special investigation to provide a unique value for each combination of w_H and V_T.

Critical IxP Moments

Critical vibratory loadings occur in the following flight conditions.

1. Transition from vertical to horizontal flight.

2. Normal Airplane flight conditions.

Moments occurring during transition flight were investigated for all angles of the thrust axis from horizontal to vertical, using the data of Reference 5 for Mp, the pitching component, and the data of References 6 and 7 for My, the yawing component. It was found that good approximation of the critical moments can be obtained by simultaneous consideration of a yawing component arising from a yawing moment coefficient C_y = .0315, and the pitching moment occurring at a thrust line angle of 75° approximated by the product of propeller thrust and an arm of .193 x propeller radius

Moments arising from normal airplane flight conditions were calculated for critical values of "A_q" per methods of References 8 and 9.

Figure 1 summarizes the most critical moments arising from consideration of the two flight conditions for the disc loadings and tip speeds investigated.

B(AF)

Both power plant weight and propeller weight are functions of the propeller blade activity factor and the number of such blades, or B(AF): the former, because of the effect of B(AF) upon HP/T and, therefore, upon power required; and the latter because of its explicit appearance in the propeller weight equation.

Selections of a "best" value of this parameter through a separate optimization procedure is possible due to the fact that for this type of mission, within the range of tip speeds and disc loadings investigated, and the practical variations of B(AF) permissible, it is found that the effect of B(AF) upon the gross weight of the aircraft is due primarily to its influence upon the empty weight of the aircraft rather than its direct effect upon fuel consumption.

Figure 2 shows the combined weights of powerplant and propeller plotted versus B(AF) for single and dual rotation propellers for several disc loadings at a representative tip speed and gross weight. Cutoffs representing minimum permissible values of B(AF) due to possibility of exceeding the allowable blade stresses (Reference 4) or maximum permissible blade stall are indicated. The B(AF) at which the combined weight is a minimum is considered the optimum. Comparison of the minimums for single and dual rotation propellers indicate the best choice.

Variation of HP/T with B(AF) was obtained from the propeller performance charts of Reference 10, and application of this optimization procedure over the complete range of disc loadings and tip speeds yielded the following selections of B(AF).

Table 1

Tip Speed	Disc Loading	B(AF)	No. Blades
800	40	450	3
	60	710	6
	80	890	6
	100	1050	6
900	40	495	3
	60	760	6
	80	780	6
	100	950	6
1000	40	520	3
	60	530	3
	80	820	6
	100	830	6

Hence, weight of the propellers for this aircraft is determined as functions of the variables of the study.

Weights of spinners, prop anti-icers, and controls are included with fixed and operational equipment in Section I.

C. Wing Weight Ratio

The wing chosen for this study is of conventional aluminum alloy sheet and stringer construction with spars located at the 15% and 50% chord. Planform and thickness taper ratios of 2:1 are assumed; the wing is equipped with leading edge slots and trailing edge simple type flaps, and is hinged at the rear spar for tilting. Although the aft 50% of the wing is not continuous across the fuselage, the structural box is not interrupted. For weight purposes a symmetrical 15% airfoil is assumed.

Critical loading conditions for the wing include +5g ultimate symmetrical maneuvering load factor, and -3.5g landing load factor in the airplane configuration with 1g airload effective per Reference 11.

The weight expressions for the wing are those previously reported in Reference 12, which was part of Progress Report Number 2 for this contract, are are not repeated here. However, changes in wing geometry necessitated changes in the values of several structural constants, and the new values are recorded below in Table 2.

In addition, in place of the assumption that W_C/W_g, the weight ratio of power package and propeller, is a constant, the calculated values of these ratios are used as they vary with disc loading and tip speed.

Table 2

Revised Structural Constants

Constant	Values Used (Dimensions Are in Feet)
k_x	3.0
α	5.26×10^{-2}
β	2.222
γ	$.02195 \times 10^{-2}$
δ	$.0182 \times 10^{-2}$
ω	$.0253 \times 10^{-2}$

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D. Fuselage Weight Ratio

Design arrangement and size of the fuselage is fixed by space requirements; hence the weight of the fuselage will vary within the range of gross weights investigated only in as much as changes in gross weight affect the general loadings.

The weight ratio for the fuselage may be conveniently expressed by a simple analytical expression as a function of gross weight based on conventional fuselage weight prediction methods (References 13 and 14).

$$\frac{W_F}{W_g} = 61.5\, W_g^{-.605}$$

E. Landing Gear Weight Ratio

Weight ratio of the landing gear is expressed by the following empirical relation (Reference 2).

$$\frac{W_{LG}}{W_g} = .045\, W_g^{.02}$$

F. Empennage Weight Ratio

Empennage weight is assumed to average 3 lb./sq. ft. of area.

Area of the horizontal and vertical tails are estimated to be (per methods of Reference 15):

$$S_{HT} = S_{wing}\,(.0217\, C - .173) + 164$$

$$S_{VT} = S_{wing}\,(.00032\, W_g^{.50} + .000136\, b) + .064\, Ap + 74.6$$

Defining:

$$b = \left[\frac{W_g\, AR}{\left(\frac{W}{S}\right)}\right]^{\frac{1}{2}} \qquad c = \left[\frac{W_g}{AR\left(\frac{W}{S}\right)}\right]^{\frac{1}{2}}$$

$$Ap = \frac{1.3\, W_g}{W_H}$$

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Total Empennage weight ratio is

$$\frac{W_{emp.}}{W_g} = 3(S_{VT} + S_{HT}) = \left\{ \left[\frac{W_g \ AR}{\left(\frac{W}{S}\right)^3}\right]^{.50} \left[.000961 \left(\frac{\frac{W}{S}}{AR}\right)^{.50} + \frac{.0651}{AR} + .000391 \right] + \frac{715}{W_g} - \frac{.520}{\frac{W}{S}} + \frac{.250}{W_H} \right\}$$

G. Supplemental Control System

The supplemental control system consisting of three lightweight turbojet engines in the aft fuselage, ducting to the tail, and a mechanism to direct the thrust in the fuselage tail cone, provides thrust required for stability and control during hover and slow speed flight.

Each engine provides 50% of the estimated required thrust so that in the event of failure of one engine, the remaining pair adequate control thrust.

From preliminary balance computations, the total thrust required is estimated to be $.049 \ W_g$ at 6000 feet on a 95°F day.

Engine specifications, based on the information in References 16 and 17, are assumed, as follows, for sea level standard day static thrust conditions.

Specific engine weight .110

SFC = 1.0 lb/T/HR (normal rated power)

Idle SFC = 1.8 lb/T/HR

Ratio-idle thrust to normal rated thrust = .160

Ratio-normal rated S.L. thrust to thrust @ 6000 ft. 95° =

1.40

Fuel for these engines is drawn from the main fuel tanks, and sufficient additional fuel is carried to operate two supplemental control engines at normal rated S.L. power plus one (the third) engine at idle for 10 minutes.

Hence, the weight ratio for the three engines, based on sea level, standard day conditions, in terms of T_e, thrust required per engine is

$$\frac{W_e}{W_g} = 3\left(.110 \frac{T_e}{W_g} \right) = .33 \frac{T_e}{W_g}$$

Fuel weight ratio for two engines operating at normal rated power, and the third standing by at idle speed is

$$\frac{W_F}{W_g} = 2(1.0) \frac{10}{60} \frac{T_e}{W_g} + 1(1.8) \frac{10}{60} \frac{.16\ T_e}{W_g} = .381 \frac{T_e}{W_g}$$

Ducting weight is estimated at 75 lbs., deflecter and controls at 125 lbs., and weight of the engine compartment including mounts, firewalls, etc. at 200 lbs.; weight of additional fuel system and tankage is estimated at 15 percent of fuel weight.

Thrust per engine at standard day sea level required to furnish the required thrust at temperature and altitude is

$$T_e = 1.4 \left(\frac{.049\ W_g}{2} \right) = .0343\ W_g$$

Hence, weight ratio of the entire system is

$$\frac{W_{scs}}{W_g} = \frac{400}{W_g} + .0264$$

H. Wing Tilting Mechanism

Weight of the wing tilting mechanism is expressed in terms of the study parameters by a term related to the loads of the system and a constant representing the weight of those items which are virtually independent of loading changes within the range of gross weights investigated.

Investigation shows good correlation is obtained from the following expression.

$$W_{TM} = 1.85\ W_J + 200$$

W_J is the weight of both jack shafts and is related to the loads and geometry as follows:

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Critical loading condition occurs during a suddenly braked landing roll with wing tilted to vertical position. Horizontal load factor $N_D = -1.75$ ultimate.

The shafts of effective column length .55 C_{root} resist a load resulting from 1.75(wing weight + power package weight + propeller weight + fuel and fuel tank weight) applied at the c.g. of the hinged mass. Total column load, from system geometry, is

$$P = 2.66 \left(\frac{W_W}{W_g} + 4 \frac{W_{PP}}{W_g} + 4 \frac{W_P}{W_g} + \frac{W_F}{W_g} + \frac{W_{FT}}{W_g} \right) W_g$$

and assuming each shaft good for 3/4 of total load to provide for asymmetry of loading, design loading for one shaft is

$$P_{cr} = \frac{3}{4} P = 2.0 \left(\frac{W_W}{W_g} + 4 \frac{W_{PP}}{W_g} + 4 \frac{W_P}{W_g} + \frac{W_F}{W_g} + \frac{W_{FT}}{W_g} \right) W_g = 2.0\, pW_g$$

W_J, weight of two threaded hollow steel shafts is approximated:

$$W_J = 2(1.5) \frac{\pi}{4} (D_o^2 - D_i^2) \left[.55\,(12) C_R \right] (.3) = 4.67\,(D_o^2 - D_i^2)\, C_R$$

$$\frac{D}{t} = 12; \text{ hence } D_i = .835\, D_o\,;\; C_{root} = 1.5\, C_{avg.} = 1.5\, C$$

so

$$W_J = 2.12\, C\, D_o^2$$

D_o^2 is defined in terms of P_{cr} on a long column

$$P_{cr} = \frac{\pi^2 EI}{L^2} = \frac{\pi^2 (3) I}{\left[12(1.5)\ .55C \right]^2} (10)^7 = 2pW_g$$

$$I_{req'd} = 6.65 C^2\, pW_g\, (10)^{-7} = \frac{\pi}{64} (D_o^4 - D_i^4)$$

$$D_o^2 = .00514\, C\, p^{\frac{1}{2}}\, W_g^{\frac{1}{2}}$$

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Hence,

$$W_J = 2.120\ (.00514\ Cp^{\frac{1}{2}}\ W_g^{\frac{1}{2}}) = .0109\ C^2 p^{\frac{1}{2}}\ W_g^{\frac{1}{2}}$$

and

$$\frac{W_{WM}}{W_g} = \frac{.0202\ C^2\ p^{\frac{1}{2}}}{W_g^{\frac{1}{2}}} + \frac{200}{W_g}$$

Noting wing chord,

$$C = \left[\frac{W_g}{AR\left(\frac{W}{S}\right)}\right]^{\frac{1}{2}}$$

$$\frac{W_{WM}}{W_g} = \frac{.0202\ W_g^{\frac{1}{2}}\ p^{\frac{1}{2}}}{\left(\frac{W}{S}\right) AR} + \frac{200}{W_g}$$

In calculating p, the fuel and fuel tank weight is necessarily assumed for first approximation

hence,
$$p = \left[\frac{W_W}{W_g} + 4\frac{W_{PP}}{W_g} + 4\frac{W_P}{W_g} + 1.15\ R_F\right]$$

I. Fixed and Operational Equipment

Weight ratio of fixed and operational equipment is estimated in the conventional manner and is the ratio with respect to gross weight of the following items.

1. Propeller Equipment Weight (Includes weight of spinners, anti-icing equipment, synchronizer, electrical beta and entc controls, etc.) . . . . . . 800 lbs.
2. Air Conditioning and Anti-icing Equipment . . . . . 500 lbs.
3. Electrical System (Empirical). . . . . . $.00375\ W_g$ + 750 lbs.
4. Instruments and Navigation Equipment . . . . 500 lbs.
5. Electronics . . . . 500 lbs.
6. Hydraulic and Pneumatic Systems . . . . . . $.005\ W_g$ + 150 lbs.

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7. Surface Controls . . . . $.010\, W_g + 600$ lbs.

8. Misc. Furnishings, Accomodations, Equipment . . . 2050 lbs.

Hence,

$$\frac{W_{FOE}}{W_g} = \frac{5300}{W_g} + .02375$$

For a 70,000 lb. aircraft, the fixed and operational equipment hence weighs approximately 7,000 lbs. exclusive of wing tilt mechanism and supplemental control system.

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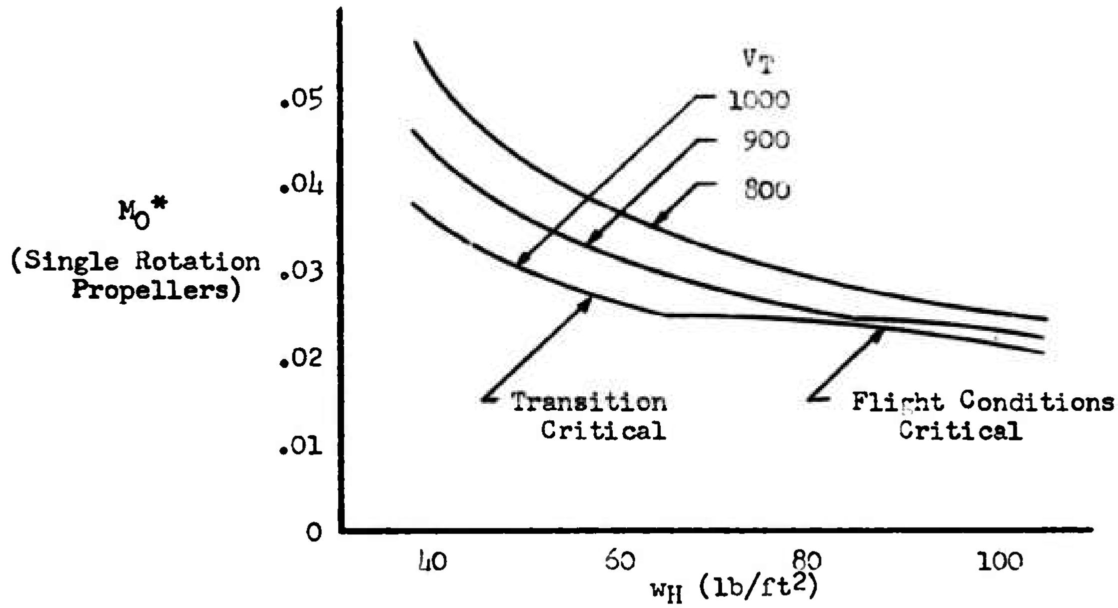
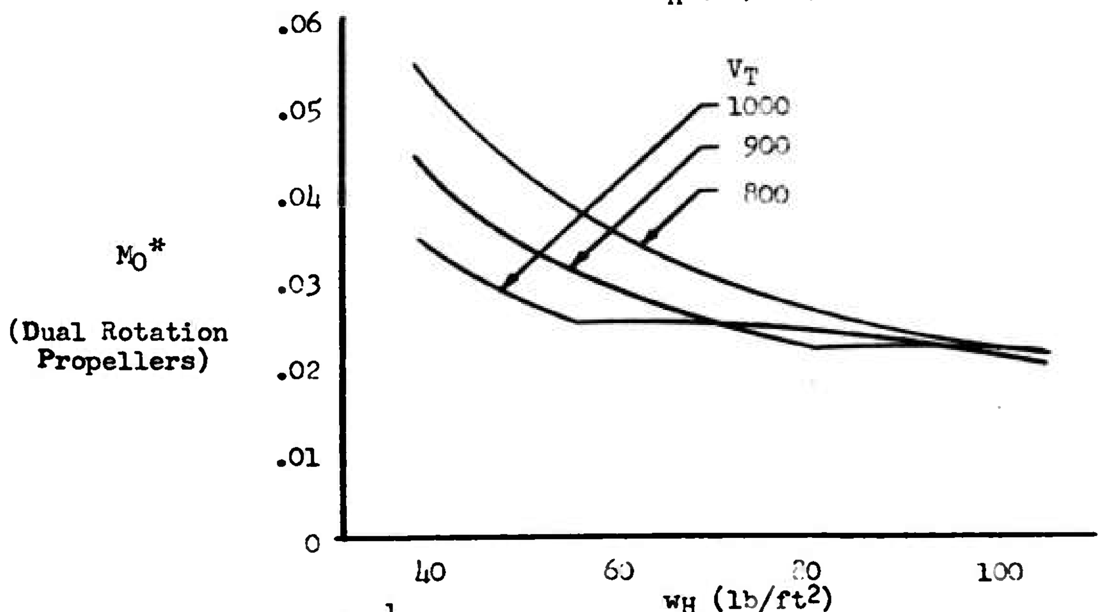
FIGURE 1.

PROPELLER IxP DESIGN MOMENTS

M_O^* @ Zero Station vs. Disc Loading.





$$M_O^* = M_O / \frac{2}{B}\left(\frac{W_g^3}{w_H}\right)^{\frac{1}{2}}$$

$$M_{O\,transition} = \left[(C_y \rho n^2 D^5)^2 + \left(\frac{r}{R} T\right)^2_{.75} D^2/4\right]^{\frac{1}{2}}$$

$$M_{O\,flight} = Aq\,(2K' AW\, D^3/K2)\cos\beta_{.7}\,(a_{.7} + 2CL_{.7}\cot\beta_{.7})$$

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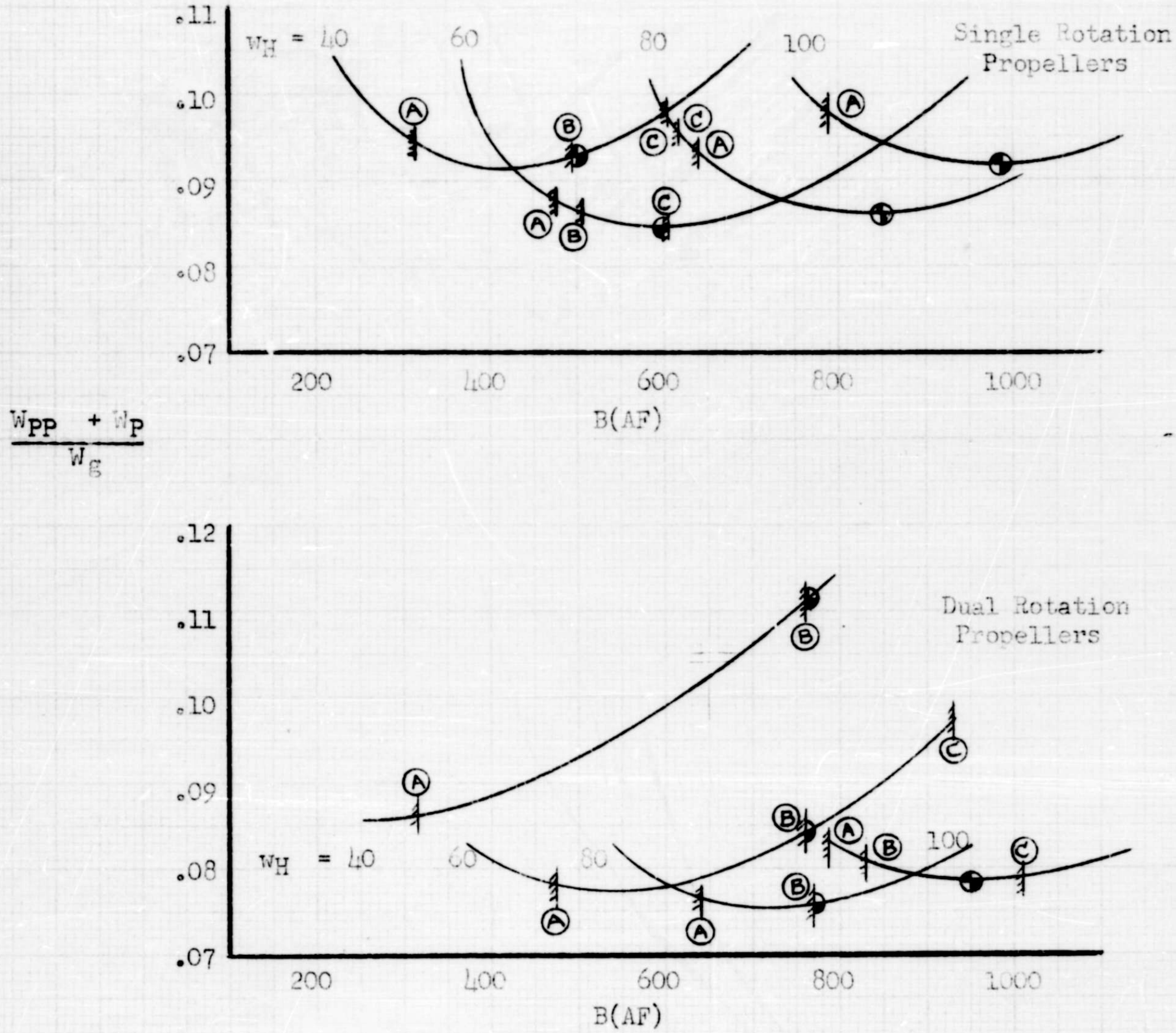
FIGURE 2.

WEIGHT RATIO OF POWER PLANT + PROPS VS. B(AF)

V_T = 900 FPS W_g = 60,000#

One Nacelle Only



LEGEND

A Maximum Allowable Stall for Control During Hover.

B Structural Limit 3 or 6 blades.

C Structural Limit 4 or 8 blades.

Optimum B(AF).

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